Combustor Design Trends For Aircraft Gas Turbine Engines

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Abstract

Advanced performance trends are surveyed for possible future gas turbine engines to power several classes of military and commercial aircraft. The resulting combustor trends are enumerated. Examples of enabling technology are given. Combustion considerations are discussed, and many commonalities between applications are discovered; fuel/air mixing and combustion dynamics emerge as topics of considerable importance.

Introduction

Technology forecasting in the aerospace field can be a hazardous past-time. For the years 1945-65, experience taught that such forecasts were generally substantially incorrect within three years of their origin, and could be seriously wrong after five years. However, these were years of rapid change involving a young product - the gas turbine engine (Hooker, 1959).

An earlier attempt at forecasting combustor technology (Sturgess, 1980), highlighted three areas that were felt likely to present a major challenge. These were combustor durability, alternative fuels, and design system changes through the introduction of computational fluid dynamics (CFD) as a tool. Some sixteen years later, actual developments in combustor durability and CFD have followed closely to the forecast; alternative fuels did not become an issue. The forecast missed the return of exhaust emissions as a serious challenge, except for smoke (in relation to alternative fuels). At a 50-50 score, this kind of performance in technology forecasting perhaps is as good as can be expected.

The nature of technology forecasting itself is subject to change, and whereas in the past technology alone was sufficient, nowadays a thorough understanding of the financial side of aerospace and its influence on technical innovation, is now a desirable attribute. In the post-Vietnam years the pace of introduction for new technology was slowed by economic considerations, and in the 1970's the aircraft gas turbine engine was being described as a "mature product" with the implication that little new could be accomplished. While this was an exaggeration, nevertheless by the 1980's an era of "incremental change" had come about. In recent years therefore, there have been more changes in the engineering process by which the engine is produced, (Gillette, 1996), than there have been in the engine. These changes have been directed towards reducing the cost and time to design and develop an engine, and towards reducing the staggering financial risk associated with a new centerline engine. Technology now has to "buy" its acceptance on an engine by offering the customer a substantial benefit of some tangible form.

Despite any inherent shortcomings, technology forecasting remains necessary and has two major functions. First, it is necessary to identify those pieces of technology that are essential to achieving further steady progress, the so-called "enabling technologies." These must be identified well in advance of their likely application, in order that they are ready for incorporation on an engine when needed. For commercial aviation the advent of full extended range (ETOPS certification) put an end to the days when a customer would...
accept "development in service." The second need for technology forecasting is to avoid being caught unprepared by paradigm-breaking events. Such an event could originate from competitive breakthroughs elsewhere, or, through the introduction of restrictive legislation, such as excessively tough noise or emissions regulations.

**Present Contribution**

The procedure for establishing goals and identifying enabling technologies is described in outline, and some of these technologies are described by means of current performance trends and Air Force initiatives. A probable paradigm-breaking event is also identified.

**Categorization of Applications**

Any engine study begins with an application. The application originates with a customer, who can be either an airline or the military. Therefore, engines for customer applications can be classified by duty-function as being either "commercial" or "military." This is illustrated in Figure 1.

![Diagram](http://example.com/diagram.png)

Fig.1: Categorization of Applications by Duty-Function

In Figure 1, engines for "transport aircraft" could see either airline or military service, since the engine design criteria will be similar. An example of this would be the Pratt & Whitney PW2037 engines in the Boeing 757 airliner, and its military counterpart the Pratt & Whitney F-117 in the McDonald-Douglas C-17 military transport. Conversely, engines for a fighter/attack aircraft clearly would be classified as only military. There are few differences in use between the commercial and military classifications of engines for rotorcraft, so these have been placed under military applications, where most of them originate. Utility aircraft are used by both commercial and military operators, and in similar fashion, the engines originate on the commercial side, where they have been placed.

Aircraft using sustained supersonic cruise, such as the proposed high speed commercial transport (HSCT) and a possible new strategic bomber for the Air Force, are specialized applications with many similarities. They have been placed separately since the engines for the HSCT are constrained by considerations of low emissions, which the engines for the bomber are not, at present.

The Air Force integrated high performance turbine engine technology (IHPTET) program is in a special category that is uniquely military. It is an engine technology demonstrator program that is intended to push the limits in every direction. As such, it represents a possible paradigm-breaking event that is perhaps only limited by costs.

**Relation Between Application and Engine Goals**

Engine goals are important since they embody the philosophy that should prevail in the design of the combustor. For example, it would not make much sense to use expensive advanced combustor liner materials, together with an elaborate cooling scheme, in an expendable engine intended for a short-range attack missile.

Most engines will have a similar list of design goals. It is the emphasis given to individual design goals that will change according to the intended application. Figure 2 illustrates this, and shows the major goals that might be applied for the different classes. For example, an engine intended for a long-range, subsonic cruise transport aircraft would have a low cruise specific fuel consumption very high on its list of goals, together with retention of this performance over long "time-on-the-wing" periods. In contrast, the engine for a utility aircraft would have reliability at the top of its list of goals. In the operating and financial circumstances that utility aircraft are used, these engines must start every time the key is turned.

**Procedure for Establishing Enabling Technologies**

Once an application is identified and the engine design goals (and philosophy) established, the enabling technologies must be determined at the component level. To
complete this step, an engine configuration and performance information are needed. Figure 3 illustrates the complete procedure.

Fig. 2: Determination of Emphasis Given to Engine Goals by Application

It will be seen in Figure 3 that considerable iteration usually takes place, and that this occurs on several levels, both internally at company level, and externally with the customer and the National Laboratories (with a role as repositories of technology). The company-internal iterations involve the rotating machinery component groups in determining the final details of the engine layout. The combustor group is not normally explicitly involved at this stage. Combustor involvement is implicit through the use of rules-based design (RBD) programs used to establish the engine configuration.

With the engine layout established and the preliminary performance (SOAP: state-of-the-art performance) table information provided, the component groups can then establish a prioritized list of design goals, consistent with the engine design goals, for the major components of the engine. This will involve some negotiation for interfacing with adjacent components. They can also commence studies to ascertain if their current design base can produce configurations that fit into the envelope allowed in the engine layout, and which satisfy the component design goals. These studies will identify any limiting technologies. Hopefully, prior use of technology forecasting should have identified these as enabling technologies, and the research and development (R&D) organizations will have appropriate solutions readily available. Figure 4 represents the acquisition time line for an advanced technology engine.

Fig. 3: Procedure for Establishing Enabling Technologies

Performance Trends

To illustrate identification of likely relevant enabling technologies for future combustors, the procedure of Figure 3 is entered at the level of engine performance.

Engine cycle performance trends for real engines at on-design conditions are complex, and depend strongly on the component losses, the amount of cooling air used in the turbines, flight conditions, and the particular design choices made, i.e. design for minimum specific fuel consumption, or, design for maximum power density. At off-design conditions, trends can be opposite to those at on-design conditions. For turbofan engines the two streams - fan and core engine - result in added complication in reaching a matched balance between the fan and core engine to achieve overall engine optimum performance aspects in a succinct manner. Actual examples
can be helpful, but such curves are invariably considered to be proprietary by the respective manufacturers.

![Graph: Acquisition of Advanced Technology for a New Engine](image)

**Fig. 4: Acquisition of Advanced Technology for a New Engine**

The general performance trends are as follows: An increase in turbine inlet temperature (TIT) always results in more thrust per unit mass of air entering the engine (specific thrust, \( F_n / \dot{m}_n \)). For a given thrust, this gives a reduction in engine weight (increase in thrust to weight ratio, \( F_n / W \)), and in frontal area, and hence, in installed drag. However, if really high temperatures are contemplated, the improved \( F_n / W \) ratio is obtained at the expense of increased fuel consumption, i.e. an increase in specific fuel consumption, SFC. At a given operating temperature, increasing the engine pressure ratio, OPR, decreases the SFC, and also specific thrust, but only slightly, as well as reducing thrust/weight ratio somewhat (through a heavier compressor). Increasing the bypass ratio (BPR) of a turbofan engine enables the entire high-pressure section (core engine) to be smaller and lighter for a given installed thrust, and gives a lower SFC. However, it increases drag through the higher frontal area of the big fan, and increases the weight through the large diameter cowl.

From the basic behavior above, it can be seen that if low SFC is an important engine design goal, (transport application), then to take advantage of all possible performance improvements, OPR, TIT and BPR should all be high. If high power density is an important engine design goal, (fighter/attack application), then OPR and TIT, should high, together with minimum frontal area and engine weight.

Company-conducted explorations of performance trends are usually confidential. However, results from a Pratt & Whitney study (Koff, 1989, 1991) provide interesting insights into some of the barriers to be overcome. For the turbo-fan engine the core is key to achieving higher performance. There are three parts to this: First, in increasing TIT as indicated above. Second, in improving rotating machinery already-high efficiencies through application of computer-designed blading to minimize secondary flow tip losses. Third, in improving the specific thrust at given TIT by minimizing wasted energy in the core engine, e.g. reducing windage losses in bearing compartments, and, in reducing air lost from the thermodynamic cycle by acting as buffer air for bearings, and as cooling air in the turbine. The power saved in this fashion can further increase propulsive efficiency by providing the energy needed to drive higher BPR fans, such as the ADF, advancedducted propulsor (Meece, 1995).

In a study for future engines trends NASA has extrapolated existing commercial engine historical design trends to beyond the year 2,000 A.D. (Taccina, 1990). For large, subsonic turbofans, these studies indicated combustor inlet pressures at cruise of 300 psia (24 atm.) with corresponding combustor inlet temperatures of 1220 F (933 K). Trends for TIT's were to values over 2540 F (1667 K). Operation at these temperature and pressure levels would confer on the engine the benefits of high thermal efficiency and low SFC. The values quoted have already been surpassed, by substantial margins, in engines presently entering service.

For commercial applications therefore, the industry (Meece, 1995) sees continued gains in performance by the year 2015 A.D., to the order of a further 25 percent reduction in SFC relative to todays levels, achieved by all the means described above, and with OPR's of 75 at BPR's up to 25, together with 100 percent increase in \( F_n / W \). The use of composite materials will play a significant and increasing role in the weight reduction, and indeed, will make the large diameter fans of the ultra-high BPR's feasible.

The U.S. Government's IHPTET initiative is aimed at taking engine technology well beyond that seen in the AFT Advanced Tactical Fighter (the F-119 engine in the F-22 aircraft). These engines experience temperatures up to 3500 F (2200 K) and pressures in excess of 40 atm. (Valenti, 1995). The IHPTET program has defined far-term goals for a variety of engines that advance in several phases. Actual performance figures are classified, but for each phase the delta's (relative to
the F-119) applied to TIT, etc., increase. Each engine component has been assigned its own goals that increase in similar fashion. Typically, for the fighter/attack application the overall engine goals, relative to the F-119/F-22 combination, are to double $F_n/W$, together with a halving of SFC. For subsonic transport applications the goal is a 30 percent decrease in SFC. The payoffs resulting from satisfaction of the initial goals are seen as:

- sustained Mach 3+ cruise capability in an F-15 size aircraft
- 100 percent range/loiter/payload increase in an F-14 size aircraft
- 100 percent range/payload increase in a CH-47 size helicopter.

For the fighter/attack application OPR's up to 100 and TTT's up to near the stoichiometric limit for hydrocarbon fuels are being considered. Since the engine goal is a high power density and low drag, BPR's remain modest. Therefore, the weight side of the $F_n/W$ balance is being forcefully addressed.

For the small engine utility/rotorcraft application, typical OPR's in the range of 25 to 50 with combustor inlet temperatures of 900 to 1450 F (756 to 1061 K), and TTT's of 2800 to 3600 (1811 to 2256 K), are being considered. These values are very aggressive compared to current performance in engines of this type.

For supersonic cruise applications studies (Smith, Jr., 1988) have generally narrowed down airliner application to Mach 2-3 cruise at 65,000 ft. (19.8 km) using kerosene-type fuel with increased thermal stability to absorb the elevated levels of fuel thermal prestressing associated with sustained supersonic cruise. The engine would be of conventional layout, and thermodynamic cycle studies have shown that raising the allowable compressor discharge temperature increases the OPR possible, and provides incentive to go up to a higher TTT. As flight Mach number is increased the thermal efficiency increases, but the maximum occurs at a lower TTT due to ram effects in the engine inlet. For Mach 2.0 cruise with 1400 F (1033 K) compressor delivery temperature, the TTT might be 2950 F (1894 K) for maximum thermal efficiency. Of course, selection of actual temperatures will represent a suitable compromise choice between the effects on efficiency, jet noise, weight, drag and exhaust emissions.

Mission requirements for a strategic bomber utilizing sustained supersonic cruise are such that novel engine configurations become necessary, with thermodynamic performance that results in engine operating conditions that are more severe than for commercial applications with supersonic cruise.

**Summary of Combustor Trends**

For the different categories of application for engines in Figure 1, regardless of mission or size, the trends for the operating conditions likely to be experienced by future combustors are the same: increased combustion pressure and increased TTT's. Table 1 summarizes likely ranges of the leading variables associated with the combustor.

| Table 1: Possible Ranges of Future Combustor Variables |
|-----------------|-----------------|-----------------|-----------------|
| Compressor exit | 0.25 - 0.35     | Mach Number     |                  |
| Inlet pressure  | 25 - 80 atm. (SLTO) | Inlet temperature | 1100 - 1600 F (867 - 1144 K) |
| Combustor configuration | annular        | Overall section pressure loss: | <3.5 % inlet total pressure |
| Compressor $L_c/H_d$ | 1.5            | Overall equivalence ratio: | 0.4 - 0.7 |
| Mean radius:    | 216 - 250 mm (regardless of core airflow) | Dome height:     | 50 - 150 mm (dependent on airflow) |
| Combustion intensity: | $(10 - 16) \times 10^4 \text{BTU/ hr. ft}^3 \text{atm.}$ | Combustion intensity: | $(10 - 16) \times 10^4 \text{BTU/ hr. ft}^3 \text{atm.}$ |

In Table 1 the ranges of combustor dimensions were arrived at as follows: Most current combustors have a burning length $L_c$, divided by the dome height $H_d$, of 2.0-2.5. With the imperative to reduce engine weight, there is a tendency to reduce $L_c$; and, in order to reduce pressure losses due to heat addition and to achieve good altitude stability, there is a tendency to increase dome height. Together, these give a reducing length to dome height ratio for future combustors.

Dome height is usually selected on the basis of combustion section airflow and minimum operating pressure. However, regardless of airflow, few combustors have achieved adequate altitude stability with $H_d < 50$ mm, due to excessive heat losses to radiation at low pressures. Thus, there is a practical minimum on dome height.

The remarkably similar values for combustor mean radius, regardless of airflow and application, arise from the obvious fact that annular combustors have the engine mainshaft passing through them, and this gives an absolute
minimum on combustor size. For large turbofans, the trend of increasing BPR results in smaller cores for a given thrust, and these cores are becoming of similar size to military engines with lower bypass ratios. For small utility and rotorcraft engines, the layout frequently involves a reverse-flow combustor that is outboard of the turbine section. Therefore, most combustors will fall in the mean radius range quoted.

It can be realized from the above that combustor designers in the near-future, are going to be faced with very similar problems regardless of the engine applications with which they are concerned.

**Examples of Some Enabling Technologies**

Some of the consequences of the performance trends and likely combustor operating conditions described will fall outside of direct combustor effects, but nevertheless will exert a considerable influence on the options available to the designer.

**Fuel pump:**

An example of this is the fuel pump. Figure 5 illustrates how the fuel pump emerges as an enabling technology. Not only is the life of the pump itself an issue, it may force a choice of fuel injector flow number on the designer. This could influence the choice of injector type and number, and hence, determine liquid fuel atomization quality and distribution within the combustor, at important engine operating points. This would be quite independent of any combustion considerations. Injector flow number is defined as follows:

\[
FN = \frac{m_f}{\sqrt{\Delta P_{\text{inj}}}}
\]

where the fuel flow rate \(m_f\) is usually given in \(\text{lbm/hr.}\), and the fuel pressure drop across the fuel injector \(\Delta P_{\text{inj}}\), is given in \(\text{p.s.i.}\).

**Super-critical fuel:**

A further example of an enabling technology is the ability to handle super-critical fuel.

JP8 fuel has a critical pressure of 42 atm., with a critical temperature of 710 F (650 K); Jet A has a critical pressure of about 21 atm. with a critical temperature of about 800 F (700 K). At sea level take-off (SLTO), current large turbofans already introduce Jet A above its critical pressure; however, the fuel maximum temperature does not normally exceed 250 F (394 K) continuous, or 350 F (450 K) with a 5-minute maximum. Evaporation of liquid fuel droplets sprayed into the combustor therefore takes place normally at the "wet-bulb" temperature of, at about 30 atm., about 563 F (568 K). This condition is sometimes referred to as "transcritical injection."

In a military aircraft, the ever-growing heat load generated by the electronics suite is now also rejected to the fuel. As examples, the F-4 aircraft had a total heat sink requirement from engine, electrical, hydraulic and crew environmental system of 6,500 BTU/min. (114.2 kW), and this was accommodated through aerodynamic cooling; however, the F-15 aircraft has a requirement of 8,000 BTU/min. (140.6 kW) and is forced to use fuel cooling. For the upcoming F-22 aircraft the heat sink requirement from all sources has risen to 18,000 BTU/min. (316.3 kW). Should this system heat rejection be combined with a sustained supersonic cruise capability, the fuel in the tanks, the heat sink, would already be quite hot, particularly towards the end of a mission when most of the fuel would have been consumed.
If the engines of the future operate at higher pressures and temperatures, the “normal” heat pick-up from the engine in delivering fuel into the combustor will increase substantially. The problem can be compounded if some of the high pressure turbine, say the inlet guide vanes, is fuel-cooled in order to enhance turbine and cycle efficiencies by reducing cooling air.

For the above, and other reasons, the ability to handle fuel supplied to the combustor in the supercritical state, will be important.

While the existence of fuel at super-critical conditions within the fuel system introduces problems of its own, such as possible flow instabilities, super-critical fuel introduced into the combustor raises the issues of achieving the required distribution of the super-critical fluid across the burning zone, fuel/air mixing, and the (unknown) combustion characteristics of fuel in this state. It does appear as though some of the physical transport properties might be continuous across the critical point, but much uncertainty remains.

Paradoxically, there may be a fuel atomization difficulty at just prior to super-critical conditions, during transcritical conditions.

The break-up processes of liquid jets and sheets in a co-flowing air-stream are similar, and that for a sheet is illustrated diagrammatically in Figure 6. The concern is with the normally fast ligament formation and breakup phase.

![Diagram of Liquid Sheet Breakup in Air](image)

For the breakup of liquid jets and sheets the fuel properties are generally characterized by the Ohnesorge Number $Z$, where

$$ Z = \frac{\mu_f}{\sqrt{\rho_f \sigma_f d_f}} $$

and the jet/sheet Reynolds Number $Re_j$, where

$$ Re_j = \frac{\rho_f (|\vec{u}_j - \vec{u}_a|) d_f}{\mu_f} $$

and subscripts ‘f’ and ‘a’ refer to liquid and air respectively; $\rho$, $\mu$, $d$, $\vec{u}$ signifying density, viscosity, diameter/thickness and velocity. Surface tension is $\sigma$. For a given value of $Z$ the Reynolds number has to be large enough for a particular fuel and jet/sheet to undergo immediate ($< 10^{-4}$ sec.) breakup.

For spherical droplets in flight, formed in the initial ligament phase, further breakup then occurs due to surface area to volume ratio effects if the Weber Number, $We$, where

$$ We = \frac{\rho_s (|\vec{u}_d - \vec{u}_a|)^2 r_d}{\sigma_f} $$

and $r_d$ is the droplet radius, exceeds a critical value of about 10-15. Such secondary breakups do not occur instantaneously, but after a specific time interval that is $1 \cdot 5.5$ times the characteristic time for spherical droplets (Faeth, 1996). The characteristic breakup time is proportional to the square root of fuel to air density ratio, the reciprocal of droplet relative velocity, and the droplet diameter, (Wu et al., 1995). During the initial phases of secondary breakup the originally spherical droplet undergoes flattening into a disk-like shape. This is due to low peripheral pressures caused by acceleration of gas over the surface. Drag coefficients for flattened droplets can be 10 - 15 times those for spherical droplets, with consequent increases in droplet velocity changes. Faeth has recognized that this secondary breakup can be rate-limiting for the atomization process.

For the limiting case of a supercritical fluid, the surface tension is zero, and both $Z$ and $We$ are infinite; atomization is therefore apparently a non-issue. An obvious difficulty arises through the relative velocity term in $Re_j$ and $We$ at sub-critical conditions but when the density ratio $\rho_s / \rho_f$ tends to unity. Under such circumstances the terminal velocity of the liquid is reached extremely quickly due to very high drag forces, and the relative velocity tends to zero before the ligament breakup process is completed (Tishkoff, 1996).
Since the critical point represents a singularity with respect to compressibility and specific heat which become infinite, and the heat of vaporization which vanishes, it is better to operate well away from this region. If the critical point has to be passed through, then this should be accomplished quickly and not near any engine steady-state operating point. Engine operation at super-critical conditions should ideally be well into the super-critical regime to avoid problems associated with the singularity. Careful engine system design will be needed to achieve this.

There is reason for concerns over atomization quality even at nominal super-critical conditions. In a super-critical combusting environment the greatly increased solubility of the gas-phase into the liquid phase can bring combustion products into the transcritical liquid phase. In addition, the droplet flattening proceeds secondary breakup results in a vortex ring formation around the periphery of the flattened droplet. The vorticity enhances entrainment of ambient fluid into the near-supercritical droplet (Mitts et al., 1996). By these means the effective critical point conditions are raised way above the nominal critical point conditions for the pure fuel. Under such circumstances surface tension does not vanish. Large droplets, as described above, could thus be left. Furthermore, the vorticity ring has high coherence, and therefore tends to persist, resisting liquid-phase breakup.

These are not the end of the problems of super-critical fuel. Since the engine has to be started, accelerated and operated at low power levels where the pressures and temperatures experienced by the fuel will below super-critical conditions, a form of "dual-fuel" injection capability will perhaps be needed.

Combustor material:

The combination of high air inlet temperatures and high combustion pressures with high TTT’s, means that extremely high heat release rates will occur in future combustors. High heat release rates will result in high heat fluxes to the combustor liners and dome. The cooling capacity of the air flowing round the combustor and used to provide liner cooling, will be reduced owing to its higher temperatures. Therefore, for a given liner material and cooling technology, an increase in coolant flux will be demanded.

If the overall bulk equivalence ratio for the combustor is high, the equilibrium levels of carbon monoxide will also be high. Any air not fully taking part in the combustion process will result in an increase in the actual burning equivalence ratio. Therefore, use of air for combustor liner and turbine inlet guide vane (IGV) platform cooling (some of which is supplied in the combustor) leads to an increase in the equilibrium carbon monoxide, and so reduces the chemical combustion efficiency. The additional large quantities of CO so-produced could emerge from the engine exhaust as a pollutant, when its heat of combustion is lost to the engine cycle. At stoichiometric conditions and relatively low operating pressures, this could amount to as much as 20 percent of the heating value of the fuel. Alternatively, if the gas temperatures remain high enough [above 2800 F (1800 K)], some of the CO could burn in the turbine and so add to it's cooling air requirement.

For commercial applications, where exhaust emissions are regulated and production of oxides of nitrogen are a special concern, precise control of fuel/air mixing to desired burning equivalence ratios is demanded. Removal from the combustion process of any air for cooling, will adversely influence the level of mixing attained and hence, will have a deleterious effect on exhaust emissions. Similarly, for low emissions designs the existence of excessive wall cooling air can result in local quenching of the fuel chemical reactions and consequently, high emissions of CO and unburned hydrocarbons (UHC’s) at low power levels.

![Fig. 7: Effect of Desired Exit Temperature Distribution on Allowable Liner Cooling Air for High Temperature Rise Combustion](image)

Although reduction in future combustor size to save engine weight will help reduce the total amount of cooling air needed, almost any cooling air is too much to meet the advanced goals. Therefore, not only will more efficient liner cooling schemes have to emerge, the availability of a suitable
advanced high-temperature liner material becomes an enabling technology.

Figure 7 is indicative of the level of material technology that will be needed. The combustor operating conditions in the illustration are modest. The calculations were made for a constant 0.1 profile factor, and the limiting overall fuel/air ratio for a 0.25 pattern factor is shown.

**Combustion Considerations**

The major enabling technologies described above establish the broad background against which the fine details of the combustion processes for future combustors have to be evolved. The details of the combustion process will be centered on two important aspects: fuel-staging, and fuel/air mixing.

**Fuel-staging:**

Fuel-staging is inconvenient, and represents increased complexity, weight and cost, together with introducing severe concerns over reliability. These problems will have to be solved since fuel-staging is necessary due to the increased range of overall fuel/air ratios (OFAR's) that will be experienced in high TIT combustor designs. Fuel-staging does have two advantages going for it: First, it is a logical extension of existing systems technology. Second, the advent of the full authority digital engine control (FADEC) makes control of staging around a complete flight envelope feasible.

If the expanded range of OFAR's is attempted in a fixed geometry combustor, either a visible exhaust smoke problem at high powers, or a loss in stability at altitude relight conditions, ensues (Bahr, 1987). Fuel-staging breaks the linear relationship between OFAR and burning zone equivalence ratio. This allows operation at the low end of the OFAR range to be designed for stability and altitude relight, and high combustion efficiency; conversely, operation at the high power end of the OFAR range can be designed for a burning zone equivalence ratio below the threshold for the onset of visible smoke. Of course, two burning zones have to be provided in some way, one for each fuel-stage.

Perfectly premixed combustion systems have very narrow stability limits that are not suitable for the wide-ranging requirements of aircraft engines. Therefore, if well-mixed systems are desired, it is necessary to provide some kind of pilot to ensure continuous and efficient burning at all operating conditions. Fuel-staging is one method of providing this. It will be shown below that such improved fuel/air mixing is necessary.

**Fuel/air mixing:**

The major drivers for commercial and military combustors are respectively, low exhaust emissions and high temperature rise. In the future these drivers may, and probably will, converge. Both will depend on improved mixing.

Exhaust emissions of oxides of nitrogen (NOx), which are regulated for commercial operations, are made up of about 90 percent by volume of nitric oxide (NO), less than 10 percent by volume of nitrogen dioxide (NO2), and 1 - 3 percent by volume of higher oxides with nitrogen tetraoxide (N2O4) being in equilibrium with NO2. The “brown tinge” sometimes presently observed in engine exhaust plumes on hot, dry days is not smoke (free carbon particles) but visible nitrous oxide (N2O) and nitrogen pentoxide (N5O5), which have a strong reddish-brown color. In the far exhaust plume, further quantities of N2O, N2O4, and nitrogen pentoxide (N5O5), which is also brown in color, can be produced from the NO and radicals present in the plume.

If very large quantities of NOx are produced by a combustor, the engine exhaust can become readily visible due to the proportionately larger quantities of the brownish gases in the plume. From a military low observables point of view, this could be a matter of growing concern.

For aviation fuels, which contain negligible chemically-bound nitrogen, NOx is produced by two mechanisms in the combustor. First, there is thermal NO via the well-known Zeldovich mechanism, which is a “time-at-temperature” process. Second, there is the so-called prompt NO (Fenimore, 1971) where in a hot, oxygen-deficient atmosphere nitrogen reacts preferentially with lower hydrocarbon fragments produced by fuel cracking, to yield NO via the amine group. In both instances the nitrogen is atmospheric. Thermal NO is generated therefore in the post-flame region, and prompt NO is generated locally, in (especially stoichiometric) flame regions.

Frequently, estimates of thermal NO are made using the partial-equilibrium approximation for atomic oxygen with the Zeldovich mechanism. Justification for this is that the overall formation rate of NO is generally slow compared to hydrocarbon oxidation reactions, and therefore, NO can be uncoupled from the fuel oxidation process and that use of equilibrium radical concentrations is appropriate.

The above approximation alone cannot account for the large rates of measured NO production. Empirical models for prompt NO, e.g. that of Moore, (Moore, 1971), often make up for most of this observed discrepancy, (Sloan, 1991) Remaining differences could be attributed to fuel/air
unmixedness effects, e.g. as in Figure 8 below, where fuel is actually reacted over a range of equivalence ratios distributed about the bulk value.

The impact on the Zeldovich mechanism of super-equilibrium radical concentrations that are present when the reacting mixture is far from equilibrium, as it is in and near flame fronts, has yet to be definitively addressed. Certainly the magnitude of the radical \( O \) concentration decreases with increasing temperature, at given equivalence ratio (Miller & Bowman, 1989, and Fenimore, 1971). Also, even though super-equilibrium effects on \( NO \) formation rates may be present over a wide temperature range, the activation energy for \( NO \) formation is so large [76 kcal/gmol - (Monat et al., 1979)] in the Zeldovich mechanism that such accelerated rates still remain low.

Support for the prompt \( NO \) mechanism as the major explanation can be provided as follows. If an overall fit of total \( NO \) data resulting from lean burning, premixed, prevaporized jet fuel, regardless of the \( NO \) source, is made with a rate expression that has the concentration dependency of a thermal \( NO \) expression, then a strong similarity exists between the resulting temperature dependency and that of the prompt \( NO \) rate expression. For example, the overall fit (Sturgess, 1991),

\[
\frac{d[NO]}{dt} = 1.2 \times 10^8 \exp(-64,000/RT)[O_2]^{1/2}[N_2] \quad \text{g mole/cm}^3\text{sec.}
\]

compared to the prompt \( NO \) expression of De Soete (De Soete, 1975)

\[
\frac{d[NO]_{\text{prompt}}}{dt} = 1.2 \times 10^7 \exp(-60,000/RT)[O_2][N_2][\text{fuel}]
\]

where exponent \( b \) varies between zero and unity.

Contributions to total \( NOx \) via the \( N_2O \) route are only significant for flame temperatures of 1500K and less, e.g. lean premixed combustors burning natural gas at modest engine operating conditions.

Therefore, for the aircraft application, it appears as though only issues associated with thermal and prompt \( NOx \) need be considered. Leaving the combustor, the balance between prompt and thermal \( NO \) depends on details of particular designs.

When the fuel and air are separately introduced into the combustor the control strategies for \( NO \) are to mix the reactants as rapidly as possible to the selected bulk equivalence ratios to avoid reaction at stoichiometric interfaces which generate large quantities of \( NO \) (Shouse et al., 1996), and to minimize the post-flame residence time (commensurate with accomplishing \( CO \) burn-up) (Sturgess et al., 1996). Most low-\( NOx \) designs involve staging of some form, and the bulk equivalence ratios may be either fuel-lean (with a pilot for stability), or, rich-then-lean. At the cost of little added complication, the pilot may be modulated to low equivalence ratios at high power to minimize its contribution to \( NOx \). The rich-then-lean choice has to deal with the difficult problem of a rapid quench between rich and lean zones to avoid the stoichiometric interface problem.

Figure 8 illustrates the realities of fuel/air mixing for a 1970's era low-emissions combustor designed to have a fuel-lean primary zone on a bulk basis, and using conventional liner pressure drops and airblast-atomizer technology.
enough to accelerate the NOx reactions. "Stirring" downstream of the primary zone cannot be guaranteed to take care of such stratification. As a consequence, measured CO levels for this combustor were very much higher than expected and measured NOx, while relatively low, was also higher than anticipated for the design equivalence ratio. The conclusion was reached that this combustor achieved its relatively low NOx at the expense of increased CO at all power levels, i.e. the combustor did not represent a true low emissions design, and was mixing-limited.

For high ITT's at high heat release rates (i.e. with smaller, weight-saving combustors), and attempted with similar mixing technology, plots like Figure 8 would exist, but at higher bulk equivalence ratios. In this case, the spread of achieved equivalence ratios would cause proportionally more of the fuel to experience local equivalence ratios greater than unity. As a consequence, CO and UHC (unburned hydrocarbons) emissions from the combustor would be high at high power conditions, and temperature rise could become mixing-limited.

It has been demonstrated (Swithenbank, 1974) that, given sufficient time, and using mechanically-generated turbulence, perfectly-stirred conditions (zero gradients of concentration and temperature) can be achieved by using a combustor pressure loss factor,

\[ \frac{\Delta P}{\dot{q}} \geq 3.0 \]

where the dynamic head \( \dot{q} \) is based on mean axial velocity upstream of the turbulence-generating device.

The combustor can be viewed as a high solidity turbulence grid enclosing the reacting flow contained within it. Therefore, Swithenbank's critical pressure loss factor should be applicable. However, experience teaches that few practical combustors are successful if the pressure loss factor is less than about 25 inlet dynamic heads, (e.g. Lefebvre, 1983). Why is there such a large discrepancy between fundamentals and practice?

Chemical reactions take place between molecules; therefore, fuel and air must be mixed on the molecular level. The length scale of turbulent eddies for molecular mixing, where introduced energy is finally produced as heat, is given by the Kolmogorov eddy length scale \( \eta \), where,

\[ \eta = \left( \frac{v^3}{\varepsilon} \right)^{1/4} \]

where \( v \) is the kinematic viscosity and \( \varepsilon \) is the dissipation rate of turbulence kinetic energy. For fully developed turbulence a Reynolds number-independent wave number \( k_d \), can be identified where dissipation reaches a maximum. Experimentally, it has been shown by Townsend and, Stewart & Townsend (Hinze, 1975) that at high Reynolds numbers most viscous dissipation occurs at a condition given by,

\[ k_d \eta = 0.2 \]

Hence, if mixing is (reasonably) equated with dissipation, the length scale for mixing \( \ell_d \), is about,

\[ \ell_d = 5 \eta \]

By these means the mixing length scales at standard atmospheric conditions can be estimated as about 0.4 mm, and at 100 atm. pressure with 1800 F (1256 K) as about \( 0.4 \times 10^{-3} \) mm.

In order to make these estimates information concerning the dissipation rates of turbulence kinetic energy was needed. Direct measurement of dissipation is not easy, and certainly no measurements at 100 atm. are known to this author. Furthermore, dissipation will vary spatially throughout a combustor. To obtain the length scale estimates given above, "typical combustor values" at lower pressures were used and were extrapolated via a suitable model, to the 100 atm. condition. From atmospheric pressure to the 100 atm. condition the dissipation estimated in this fashion varied by some orders of magnitude. Therefore, the length scale values given should be treated with caution until actual measurements become available, and should be viewed as only representing the likely ranges to be encountered. However, the turbulence assumptions made at the 100 atm. condition place such a combustor on a Borghi plot consistent with other combustors of Sattelmayer and of Correa at pressures from 1 to 15 atm., Sturgess, 1996.

Consider now the typical conventional combustor of annular form. The circumferential spacing between fuel injectors is about 1.0 dome height, the first row of air addition ports are placed between 0.5 and 1.0 dome heights downstream from the dome, with about half-a-dome height spacing between jets, and, the combustor length is 2-3 dome heights, of which the intermediate zone is about half-a-dome height in length, as is the dilution zone. Partial modeling has demonstrated that to first order, the gross flow field in a combustor is determined by the confining geometry (Spalding, 1956). The characteristic length for a conventional combustor can therefore be taken as the dome height, \( H_d \). From Table 1, dependent on airflow,

\[ 50 < H_d < 150 \text{ mm} \]
Given the vertical cross-sectional symmetry, the major flow structures established in the combustor will hence have a linear dimension on the order of 25 - 75 mm.

When the typical flow structure dimension in a conventional combustor is compared to the length scale of the eddies for turbulent mixing, the dynamic range is from about 60 at low pressures to about 190 x 10^3 at high pressures. Therefore, although gross Reynolds numbers might suggest otherwise, the turbulence, on a bulk basis, is not fully developed throughout the combustor. Intense mixing thus only occurs in a relatively few shear layers of limited extent, that exist around the peripheries of the major flow structures present in the combustor. For example, the shear layers between the liquid fuel sheet and the air jets associated with it in airblast atomizing fuel injectors, the shear layers between the fuel/air spray from the injectors and the dome recirculation zones, and the shear layers between the transverse air jets and combustion gases moving around them. The majority of the energy introduced into the combustor by the pressure drop across the liners goes, not into direct mixing, but into bulk mass transport over the relatively large distances between flow features. This explains the difference between theory and experience with respect to combustor pressure loss factor.

With the present uncertainties described above with respect to dissipation rate at very high pressures, the absolute magnitude of the mixing length scales may be inaccurate. However, the relative magnitudes with respect to the combustor flow structure sizes certainly are in the correct range. Therefore the conclusions drawn are valid.

If improved fuel/air mixing is required due to the needs to control emissions and to achieve high heat release rates, then combustion section pressure loss must be utilized much more effectively. For cycle performance reasons it cannot be increased. More effective use of pressure loss implies. 1.) a reduction in the combustor characteristic length scale by at least an order of magnitude, and, 2.) an increase by a factor of at least two, in the number of shear layers existing and active in the combustor.

The inference from these two changes, and the implication from partial modeling (Spalding, 1956), is that fairly radical changes in combustor geometry will be needed. The geometric layout of gas turbine combustors has existed essentially unchanged since the days of Whittle and von Ohain. Therefore, the changes suggested may fairly be described as "paradigm-breaking." However, if mixing is made more effective, it might be possible to contribute towards achieving two additional important engine goals: 1.) a reduction in combustor pressure loss, and, 2.) a reduction in section length. Such reductions would help SFC and $F_{W, in}$ respectively.

Emissions:

The well-known adverse atmospheric and environmental effects of gas turbine emissions, and especially NOx, have led to increasingly severe regulatory action for commercial aircraft operations (Segalman et. al, 1994). The direct concern over NOx - emissions for military visibility reasons is described above. There is an additional cause for concern. NO2 has lethal toxic potency, as does NO but only at one-fifth the potency of NO2 (Lilley, 1996). Fatality with NO2 occurs within 10 minutes at exposure levels of 200 ppm. As a pulmonary irritant, prolonged exposure to NOx at low concentrations has also been identified as a cause of emphysema. Now, in addition to normal airfield taxi-take-off-landing operations, a modern military warplane usually spends some 30-45 minutes on the ground, prior to a mission, at engine idle power while the electronics suite is aligned and calibrated. Therefore, the potential hazard to flight-line crews during both routine and extended operations can be appreciated.

Improved fuel/air mixing has been identified as a necessary control parameter for emissions. However, it appears that fairly drastic design changes will be necessary to achieve significant improvements in the direct, in situ, mixing required to avoid the twin dangers of auto-ignition and flash-back associated with premixing. Therefore, it is desirable to assess the impact of advanced engine performance (as typified by the IHPTET effort) on NOx emissions.

Figure 9 displays projections of the emission indices of NOx (as NO2) as a function of pressure up to 100 atm., at a fixed overall equivalence ratio of 0.5, for a number of different combustors. Combustor inlet temperature was varied with the pressure along an assumed compression characteristic. The combustors shown represent, from top to bottom, a single-stage conventional combustor, an axially-staged combustor utilizing otherwise conventional technology (Segalman et. al, 1994), a fuel-staged, increased-mixing combustor, and a single-stage, ideal, prevaporized, premixed combustor with all of the combustion air introduced through the dome. The "ideal" combustor is of course, impractical, but serves as a limit on what is attainable. All of the combustors assumed use of an advanced liner material/cooling scheme that utilized a maximum of 15.5 percent of the total combustor air for cooling and profile-trim purposes; this air was taken as not being available for combustion purposes. The (hot) residence time in all combustors was fixed at 2.7 ms. All NOx values are corrected to standard humidity.
In each case considered, except for the "ideal," a calibrated "flow model" for that combustor configuration was exercised over the range of operating conditions, and involved solving the same detailed chemical kinetic reaction mechanism on the appropriate flowfield. Both thermal and prompt NOx were included. This procedure is considered to be more accurate than just extrapolating a simple NOx correlation. It is important to note that a completely independent assessment (by methods unknown to the present author) of the fuel-staged, increased-mixing combustor over this pressure range produced results virtually identical in slope and almost equal in magnitude to the present calculations. However, it is naive not to accept that some level of uncertainty is involved over the wide range of pressures covered.

The curve for the "ideal" low-NOx combustor shows quite high levels of NOx being produced at 100 atm. relative to levels produced in current engines. Therefore the term "low-NOx" as applied to combustors for future engines is a misnomer; perhaps "minimized NOx" or "contained NOx" might be better. It should be remembered that Figure 9 is expressed in g NOx per kg fuel burned. Use of increased pressure would reduce the amount of fuel burned for a given mission. This might, in turn, give a reduction in the absolute mass of NOx from integration over an operating envelope. Therefore, not only are the absolute values of NOx in Figure 9 important, but also the gradients.

It can be seen that the fuel-staged, but otherwise conventional, ASC combustor (Segalman et al., 1994), does very well on containing NOx production compared to the conventional combustor. Figure 10 illustrates the NOx reduction that is achievable just due to staging, (Segalman et al., 1993).

For the ideal combustor, the calibrated lean, premixed, prevaporized, jet-fuel NOx rate equation given above (Sturgess, 1991) was integrated directly.

The calculated NOx from the conventional combustor was found to increase rapidly to very high levels with increasing pressure. These levels reached equilibrium values about 60 atmospheres. Quite clearly, the environmental impacts, visibility issues and potential health effects will most likely preclude the use of conventional combustor technology in advanced engines of the future.
The NOx characteristic for the fuel-staged, increased-mixing combustor falls between the ASC and the ideal cases, being closer to the former than the latter. However, it does represent an improvement over existing technology. The improvement over the ASC is primarily attributable to the improved fuel/air mixing associated with a reduced length-scale in the main stage. It represents the first step in what might be possible if the design paradigm is broken, as described above.

The curve is constructed with an extrapolation (a very large extrapolation, admittedly) using modeling, of experimental results obtained in a rig demonstrator at atmospheric pressure. Whether it is possible to develop such a combustor into an "engine-worthy" design, and while doing so retain the reduced emissions or even improve on them, remains to be seen. Note also that the residence time is the same as that for all of the combustors. With improved mixing, it might be possible to reduce the residence time. Such a reduction would further lower the thermal NOx produced.

Extending the analysis in parametric form for overall equivalence ratio revealed quite severe increases in NOx levels, with even the "ideal" combustor producing 200 g/kg at 100 atm. for 0.7 equivalence ratio. To restore the NOx of the "ideal" combustor at 100 atm. and 0.7 equivalence ratio to the previous 100 g/kg, for 0.5 equivalence ratio at this pressure, it was necessary to reduce the residence time from 2.7 ms to under 1.5 ms.

It will be observed in Figure 9 that the calculated gradients of NOx with increasing pressure for the two fuel-staged combustors and the "ideal" combustor do not vary very much over the pressure range, and agree reasonably closely with each other.

In general, the pressure exponent for NOx depends on which of the NOx generation mechanisms dominates in a combustor, and, for each mechanism, can vary significantly with local equivalence ratio, i.e. the dependency is determined by both equivalence ratio and adiabatic flame temperature. The behavior is therefore complex. An example of this complexity is illustrated in Figure 11 for experimental lean methane/air mixtures at 322K inlet temperature, burning on a perforated plate flameholder at 3 - 10 atmospheres, and calculations made over a wider range of conditions up to inlet temperatures of 728K and pressures of 14.6 atmospheres (Sturgess, 1992). In the figure, the pressure exponent is plotted against adiabatic flame temperature. The generality of such calculations will depend, of course, on the accuracy of the reaction mechanisms used and the reliability of the associated rate constants.

While the experimental data in Fig. 11 is very limited, the close agreement with the calculations in this region can be noted. It is also well-known that for adiabatic flame temperatures of 1500K and less for methane/air systems, N2O is the dominant pathway to NOx.

Accurate representation of the pressure dependency of NOx is clearly important when making calculations at greatly elevated pressures, as in Fig. 9.

The pressure dependencies of the fuel-staged combustors shown in Figure 9, when these combustors are operated over the same range of conditions, therefore depends on the details of the emissions-control strategy used, and also, on the degree of fuel/air mixing achieved in each design. Clearly, these may be judged to be somewhat similar for the cases shown in the study.

![Fig. 11: Provisional Pressure Dependency for NOx](http://proceedings.asmedigitalcollection.asme.org/)

Figure 12 illustrates why the "ideal" combustor will remain an ideal, and why it will be difficult to achieve the pre-combustion improved fuel/air mixing essential to lower NOx generation. The figure presents the variation of estimated ignition delay times along the compression characteristic for perfect premixing and for the relatively poor mixing rates typical of current systems. The estimated mixing and ignition delays are for vaporized JP8/Jet A to simulate fuel injected in the super-critical state. The ignition delays were found from a...
Relatively poor mixing MOS tk, S—et—Mixing time — 4. • Ignition delay. premixed systems — re •Autoignition time, concurrent mixing data correlation of JP5/Jet A/JP4/Avtur etc., and that included data up to 60 atm. pressure (Sturgess, 1979).

<table>
<thead>
<tr>
<th>Time, ms</th>
<th>Mixing time</th>
<th>Ignition delay, premixed systems</th>
<th>Autoignition time, concurrent mixing</th>
</tr>
</thead>
<tbody>
<tr>
<td>0.001</td>
<td>0.1</td>
<td>1.0</td>
<td>10.00</td>
</tr>
<tr>
<td>0.01</td>
<td>0.001</td>
<td>0.5</td>
<td>5.00</td>
</tr>
<tr>
<td>0.1</td>
<td>0.01</td>
<td>0.001</td>
<td>0.10</td>
</tr>
</tbody>
</table>

**Fig. 12: Influence of Compression Characteristic on Autoignition Behavior for Current and Perfect Mixing Systems**

Ignition delay times for perfect premixing are less than 0.1 ms for pressures above 30 atm. For current systems it can be seen that the estimated autoignition occurs before the estimated mixing is complete, and this concurrent behavior is to be expected. Ignition delays are less than 1 ms.

The implications of the calculations are as follows: 1.) perfect premixing will be impossible at elevated pressures and the temperatures that go with them, 2.) design of any kind of practical fuel/air mixer to achieve some, but less than perfect, mixing prior to combustion will be exceedingly difficult, since flow wakes and separations must be avoided under all circumstances, and, 3.) direct injection of fuel, but with wide dispersion to improve presentation to combustion air, appears as a lower-risk approach.

**Combustion dynamics:**

The trend for the future would appear to be towards improved mixing. It is well-known that well-mixed combustion systems can be notoriously susceptible to nonstationary behavior of the mean flow field, with the possibility of acoustic coupling. The term "combustion dynamics" is coming into use to describe this type of behavior. The nonstationary behavior situation is exacerbated when the well-mixed combustor is operated lean as an emissions control strategy. High combustion pressures further increase the tendency towards mean flow instability.

If an engine is designed with high turndown ratios and ultra-high combustion pressures, and the fuel pump enabling technology is not adequate, use of high flow number fuel injectors may be forced on the designer. One implication of high flow number fuel injectors is that at low engine power levels, the fuel system at the combustor does not "run full." This means that the fuel supply is "acoustically open" to the combustor. In a well-mixed, fuel-lean combustor that is dynamically active, a driven oscillation can then be established, especially during low power level engine transients.

The consequences of such combustion dynamics can range from increased engine noise (passenger annoyance, regulation violations), increased heat transfer in the combustor, ingestion of hot partial-combustion products into fuel injectors (coking problem), and, engine component (pipework usually) resonances.

This is a phenomenon that is present to some degree in many of today's conventional combustors, and is commonly referred to as "rumble." Frequencies range from about 50 to 500 hertz, and peak-to-peak amplitudes can be rather high. In these cases it is either ignored or "engineered around." Neither of these treatments are expected to be adequate if the combustor changes described above come into production. It is an area that is expected to assume increasing importance in these future combustors. Unfortunately, our understanding of, and our ability to calculate, combustion dynamics is rudimentary at best. The possibility of solving these difficulties, or even exploiting them to advantage, by means of active combustion control (ACC) is an intriguing possibility.

**Conclusions**

Many important issues that are relevant to combustion systems for future aircraft gas turbine engines have been touched upon. In such a presentation as this the treatment of individual issues can only be cursory. Much is still unknown in many of the topics highlighted, and research work is proceeding in these areas. It is most likely that some of the issues presented will turn out not to be major blockages to progress, while others will surface that are not presently mentioned. Variable geometry, active combustion control, endothermic fuels, and the use of micro-electronic and mechanical systems (MEMS) technology are not discussed.
Their omission is deliberate. These are dynamic areas of great potential, subject to on-going research and development, and as such, the associated technology is sensitive.

From the areas discussed the following general conclusions can be drawn:

1. Future engine layouts and performance will cause combustors for many applications to become similar in general size. Common design problems and solutions will come about.

2. Combustor operating conditions will continue to increase in severity through increased operating pressure, air inlet temperature and temperature rise.

3. Major enabling technologies have been identified as:
   a) an ultra-high pressure fuel pump capable of handling hot fuel,
   b) the ability to deliver and burn super-critical fuel,
   c) high temperature liner materials.

4. Major design challenges for the combustor are:
   a) increased space heating rates with no increase in pressure drop,
   b) containing exhaust emissions as operating conditions increase.

5. Significant changes to the combustor will include:
   a) certainly, fuel staging for high turndown ratios,
   b) probably, geometry changes for enhanced fuel/air mixing,
   c) possibly, active combustion control for engine health maintenance, reduced noise, and perhaps, enhanced fuel/air mixing.

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